NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE 3620

THE DESIGN OF A MINIATURE SOLID-PROPELLANT ROCKET

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SUMMARY

A miniature rocket motor was designed and developed to produce 3 ounces of thrust for a duration of 2 seconds. The rocket is simply designed, safe to operate, easily handled, and gives reproducible performance. Standard solid-propellant-rocket design techniques were found to be not wholly applicable to the design of miniature rockets because of excessive heat losses.

INTRODUCTION

Free-flying dynamic scale models are used frequen[illegible] [illegible]uring aerodynamic research data. These models generally use [illegible] aerodynamic surfaces in order to furnish disturbing or restor[illegible] [illegible]orces and moments for purposes of investigating the response of the model. Often, these aerodynamic surfaces are in unsteady or unknown flow fields; consequently, the magnitudes of forces or moments being applied are not accurately known. The use of small rockets to produce the required disturbing forces or moments minimizes these difficulties. Rocket motors have been used with good results to disturb the flight of rocket-powered research models in pitch and yaw. Recently, a requirement for the disturbance of a free-spinning model in the Langley 20-foot free-spinning tunnel resulted in the development of a miniature rocket producing 3 ounces of thrust for 2 seconds. (See ref. 1.) Inasmuch as presently known solid-propellant-rocket techniques were inadequate for the design of miniature rockets, the engineering methods necessary to produce the desired characteristics and the steps of research and design necessary to fabricate such a rocket are presented herein.

SYMBOLS

A area, sq in.

C_d discharge coefficient, per sec

C_F thrust coefficient

d	diameter, in.
F	thrust, oz or lb
I_{sp}	specific impulse, lb-sec/lb or sec
I_t	total impulse, oz-sec or lb-sec
K	restriction ratio, S/A_t
p	static pressure, lb/sq in. abs
r	propellant burning rate, in./sec
S	propellant burning surface, sq in.
t	time, sec
w	mass discharge rate, lb/sec
γ	ratio of specific heats

Subscripts:

a	atmospheric
c	rocket combustion chamber
e	exit
t	rocket throat

DESIGN PROCEDURE

According to the design requirements specified for the miniature rocket of reference 1, the rocket motor was to produce 3 ounces of thrust for a duration of 2 seconds and was to weigh less than 20 grams. No commercially manufactured rocket motor was available or could be modified to meet these requirements. The weight requirement stipulated a miniature rocket having a propellant that would operate at low pressures in order to keep the weight of the metal parts to a minimum. Cordite SU/K, a double-base extruded propellant of British origin, was selected because of its desirable ballistic properties at relatively low operating pressures and because it could be easily machined. A design chamber pressure of 315 lb/sq in. abs was chosen because past experience had shown that rocket motors using Cordite SU/K propellant will sometimes enter an unstable burning period referred to as "chuffing" if lower operating pressures occur. (See ref. 2.)

An end-burning charge in the shape of a circular cylinder was chosen to study the parameters of surface area and length necessary to produce the desired results and to give a reasonable shape to the rocket motor. The propellant charge was to be inhibited on all lateral surfaces and at one end in order to produce end burning along the axis of the propellant. A simple orifice-type nozzle with an area ratio A_e/A_t of 1 was selected. For preliminary design purposes standard internal-ballistic equations were used to determine the initial propellant size. These equations may be found in reference 2. Since the ratio of nozzle exit area to nozzle throat area was 1, the exit pressure p_e was equal to the throat pressure p_t. The exit pressure p_e was computed from the following equation, by using the assumed value of chamber pressure p_c of 315 lb/sq in. abs:

$$p_e = p_t = p_c \left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma}{\gamma-1}} = 315 \times 0.555 = 175 \text{ lb/sq in. abs}$$

where the value of γ was determined to be 1.25 for Cordite SU/K propellant. Since the values for A_e/A_t, p_c, p_e, γ, and required thrust are known, a value for thrust coefficient C_F and required nozzle throat area A_t may be calculated as follows:

$$C_F = \sqrt{\frac{2\gamma^2}{\gamma - 1}\left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma+1}{\gamma-1}}\left[1 - \left(\frac{p_e}{p_c}\right)^{\frac{\gamma-1}{\gamma}}\right]} + \left(\frac{p_e - p_a}{p_c}\right)\left(\frac{A_e}{A_t}\right) = 1.20$$

$$A_t = \frac{F}{C_F p_c} = \frac{0.1875}{378} = 0.000496 \text{ sq in.}$$

The propellant burning area S is determined from the restriction ratio $K = S/A_t$. The restriction ratio required is a function of design chamber pressure and must be determined from experimental data. Figure 1 shows the variation of restriction ratio K and burning rate r with chamber pressure p_c for full-scale rocket firings of Cordite SU/K propellant. These data were obtained from previous static firing tests of many full-scale Cordite rocket motors. Figure 1 shows that, for a design chamber pressure p_c of 315 lb/sq in. abs, the restriction ratio required is 207. The propellant burning surface area, therefore, must be 0.102 square inch. Figure 1 also shows the burning rate to be 0.227 inch per second. Therefore, for a 2-second duration, the propellant length must be 0.454 inch. By using the above calculated size, a charge in the shape of a right circular cylinder was machined from propellant stock.

An inhibiting material was needed that would withstand the high temperature and erosive effects of the propellant gas and would cause the propellant to burn progressively rearward in order to maintain a constant burning area and the desired burning rate. This in turn would cause the pressure level to remain constant and thereby produce the required 3 ounces of thrust. Also, the inhibiting material should not cause a change in the ballistic properties of the propellant. The propellant was wrapped with several layers of cellulose acetate tape as an inhibitor for initial tests.

An igniter was made by utilizing a commercially manufactured match-type squib which consists of a cardboard tube containing two insulated wires with a match head at the end and having a sulfurized compound as a filler and insulating material. The squib was held in place by a metal nut in the igniter holder which was placed at right angles to the propellant chamber. The original test rocket was constructed of SAE 1020 steel with a removable head cap.

TEST PROCEDURE AND ANALYSIS OF RESULTS

Instrumentation

The thrust stand used in the experimental rocket tests consisted of a deflected table which transmitted the thrust load to a small strain-gage force pickup. An electrical strain-gage pressure pickup was connected to a pressure probe installed in the rocket chamber wall. These instruments, in conjunction with a recording oscillograph, gave continuous records of thrust and pressure as a function of time. A miniature rocket is shown mounted on the thrust stand in figure 2.

Results of Preliminary Tests

Figure 3 shows a thrust-time history obtained from one of the initial miniature-rocket-motor tests. Several static firings of the rocket motor revealed the following conditions: improper ignition characteristics, low thrust and pressure particularly during initial burning, and unsatisfactory inhibiting of the propellant. These conditions were attributed to a lack of igniter flammability, severe heat losses immediately after ignition, and insufficient propellant burning surface necessary to sustain 3 ounces of thrust.

In order to increase the flammability of the igniter, the match-head end of the squib tube was filled with a fine-grained black powder cemented in place with an acetone adhesive. In order to improve the inhibitor and to maintain a constant burning rate, a method was devised to fabricate a phenolic shell, which was then bonded to the propellant by a coating of an epoxy adhesive on the inside of the shell and lateral propellant surface.

A study of the test results revealed that the propellant experienced a severe heat loss to the chamber walls immediately after ignition. This heat loss in miniature rockets is much greater than the heat loss occurring in larger rockets because the propellant gives off heat to the chamber walls at a rate proportional to the cube of the scale factor, and the metal parts absorb the heat at a rate proportional to the square of the scale factor.

In order to compensate for the heat loss the propellant grain was modified by the addition of an uninhibited cylindrical protrusion 0.25 inch in diameter and 0.05 inch long. This protrusion also compensated for the low thrust immediately after ignition. Static firing tests using the new charge design with increased surface area were made. The results of these tests disclosed that the additional surface area compensated for the heat loss and maintained the required thrust. However, this also resulted in a higher chamber pressure (353 lb/sq in. abs) and a higher propellant burning rate. Consequently, it was also necessary to increase the length of the propellant charge in order to achieve a 2-second burning time. A thrust-time curve obtained from the static firing of this compensated propellant design is shown by the dashed-line curve in figure 3. The results show that solid-propellant-rocket design techniques, taken from full-scale firings, can be applied in the initial design of miniature rockets but compensatory methods must be utilized because of excessive heat losses encountered.

Results of Tests of Revised Rocket Design

Many static firings of the revised rocket design were made to determine its ballistic characteristics and serviceability. Figure 4 shows thrust-time curves for several static firings of the miniature rocket. These data show the rocket to have close repeatability of performance.

The igniter design proved to be satisfactory. However, after several firings a phenolic nut instead of a metal nut was fabricated to hold the igniter squib because this type of construction provided an inexpensive reliable igniter that affords a pressure-relief device in the event the chamber pressure becomes high enough to cause a possible rupture of the case and danger to the users of the rocket or damage to the models. The phenolic nut was sealed in place with the same epoxy adhesive used as the inhibiting material. The inhibitor was completely satisfactory in that it caused no variation in the ballistic properties of the propellant. In order to simplify the design and insure positioning of the propellant in the chamber, the head cap was made an integral part of the inhibitor shell; thus, the design provides an inexpensive expendable unit consisting of propellant, inhibitor, and head cap all in one piece. Repeated static firings of the reusable rocket case revealed a loss of thrust caused by an enlarged orifice created by erosive propellant gas flow during burning. In order to avoid this problem the rocket case was modified to incorporate a replaceable nozzle block as shown by a cutaway sketch of the miniature rocket and components in figure 5. The miniature rocket incorporating these modifications proved to be simply designed, safe to operate, easily handled, and to give reproducible performance. Performance characteristics of the miniature rocket are compiled in table 1.

CONCLUDING REMARKS

A miniature rocket motor was designed and developed to produce 3 ounces of thrust for a duration of 2 seconds. Standard solid-propellant-rocket design techniques were found to be inadequate for the design of a miniature rocket because of excessive heat losses.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., November 25, 1955.

REFERENCES

1. Burk, Sanger M., Jr., and Healy, Frederick M.: Comparison of Model and Full-Scale Spin Recoveries Obtained by Use of Rockets. NACA TN 3068, 1954.

2. Wimpress, R. N.: Internal Ballistics of Solid-Fuel Rockets. McGraw-Hill Book Co., Inc., 1950.

TABLE I.- WEIGHTS AND PERFORMANCE CHARACTERISTICS OF

THE MINIATURE ROCKET MOTOR

Weight of rocket motor components in grams:	
Rocket motor case	14.17
Inhibited propellant and shell	2.44
Cordite SU/K propellant	1.16
Igniter assembly	1.95
Rocket motor and components	18.56
Experimentally determined performance parameters (average values at 3 ounces of thrust):	
Nozzle discharge coefficient, C_d, per sec	0.00745
Thrust coefficient, C_F	1.08
Specific impulse, I_{sp}, sec	145
Total impulse, I_t, oz-sec	6.50
Discharge rate, w, lb/sec	0.0013
Adiabatic flame temperature, $^{\circ}F$	3,800
Chamber pressure, p_c, lb/sq in. abs	353
Restriction ratio, K	205
Nozzle exit pressure, p_e, lb/sq in. abs	196
Burning time, t, sec	2

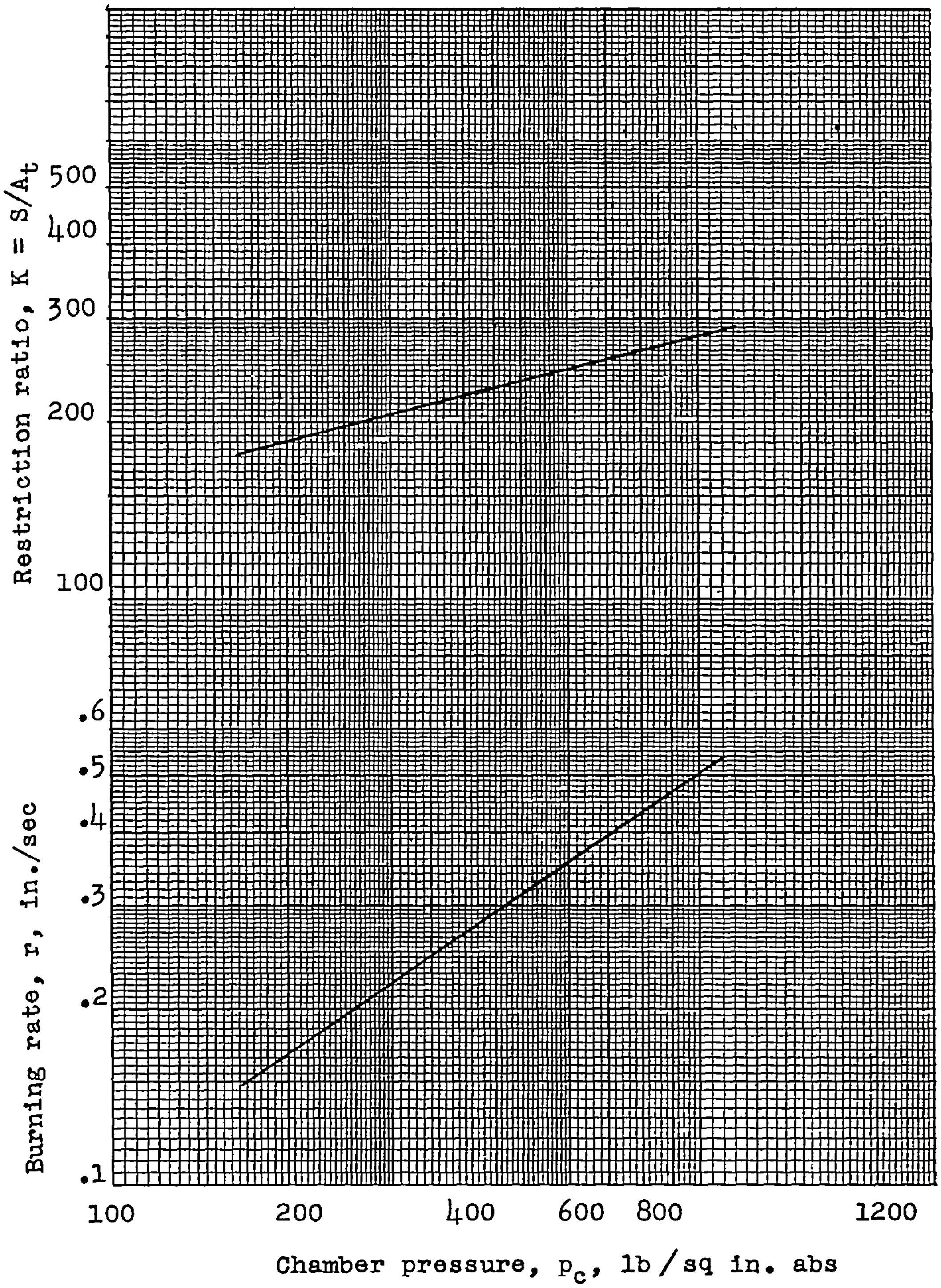

Figure 1.- Variation of burning rate and restriction ratio with chamber pressure for full-scale rocket-motor firings using British Cordite SU/K propellant.

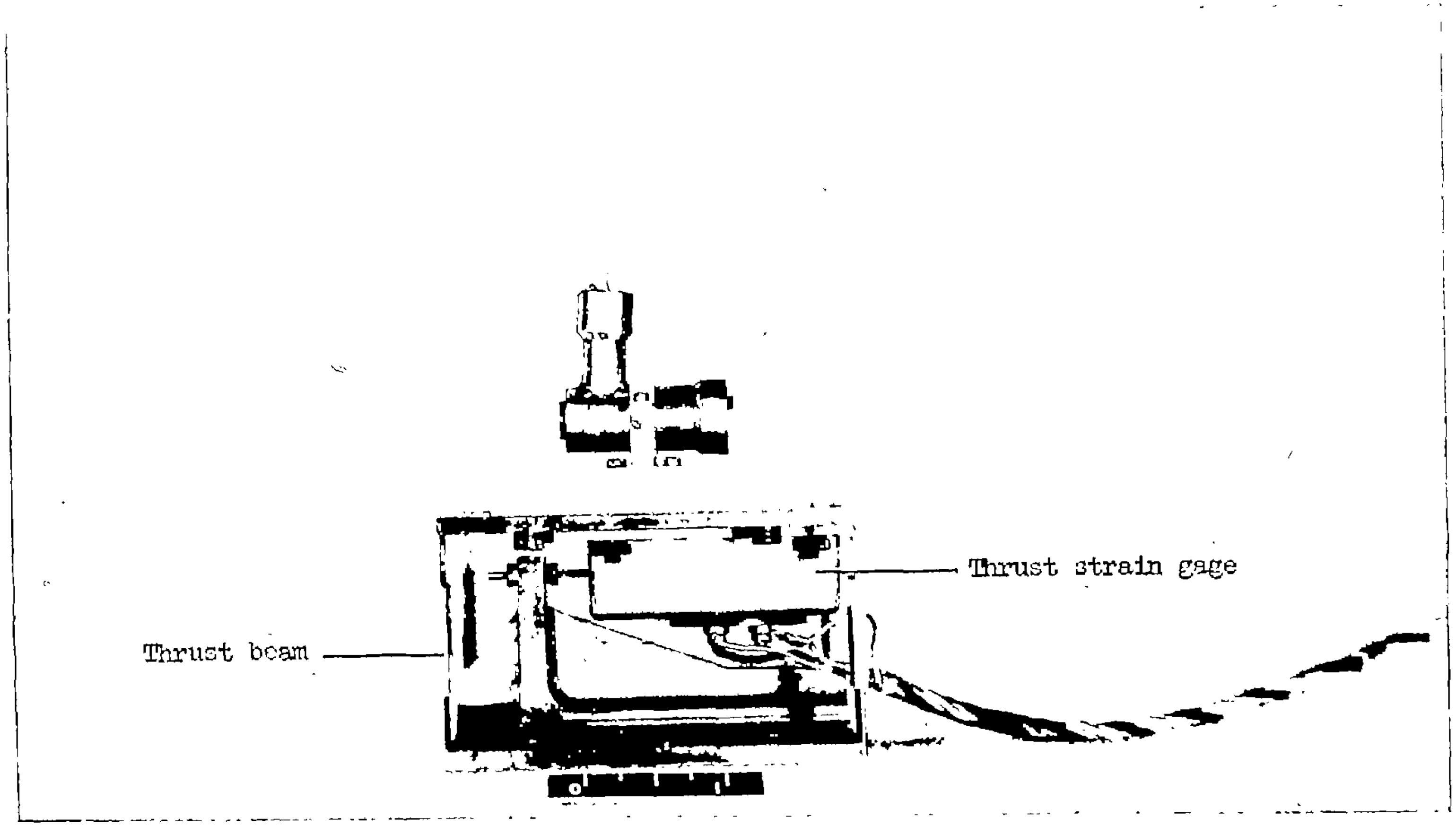

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Figure 2.- Static thrust stand and strain gage with the miniature rocket mounted.

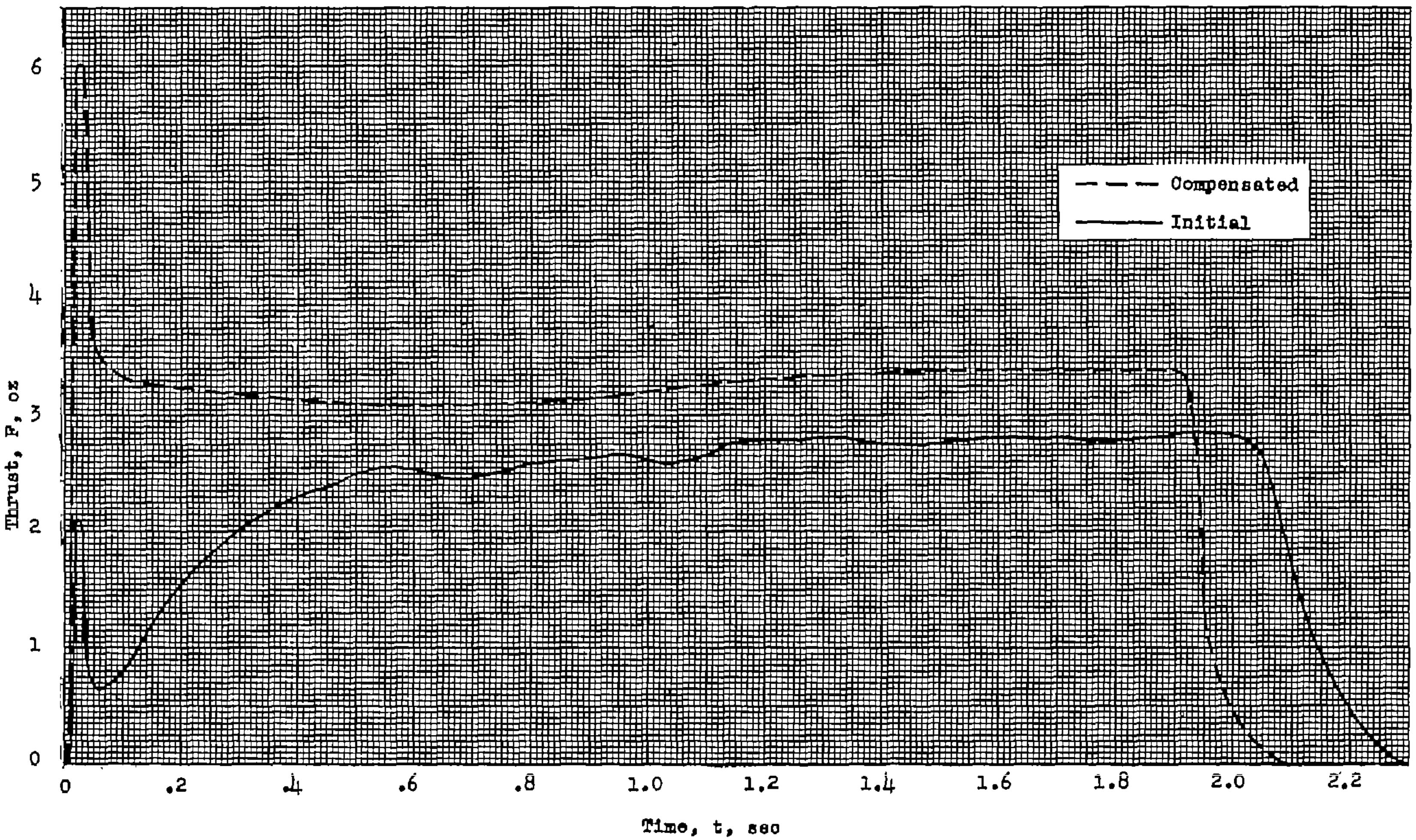

Figure 3.- Variation of thrust with time for two propellant designs.

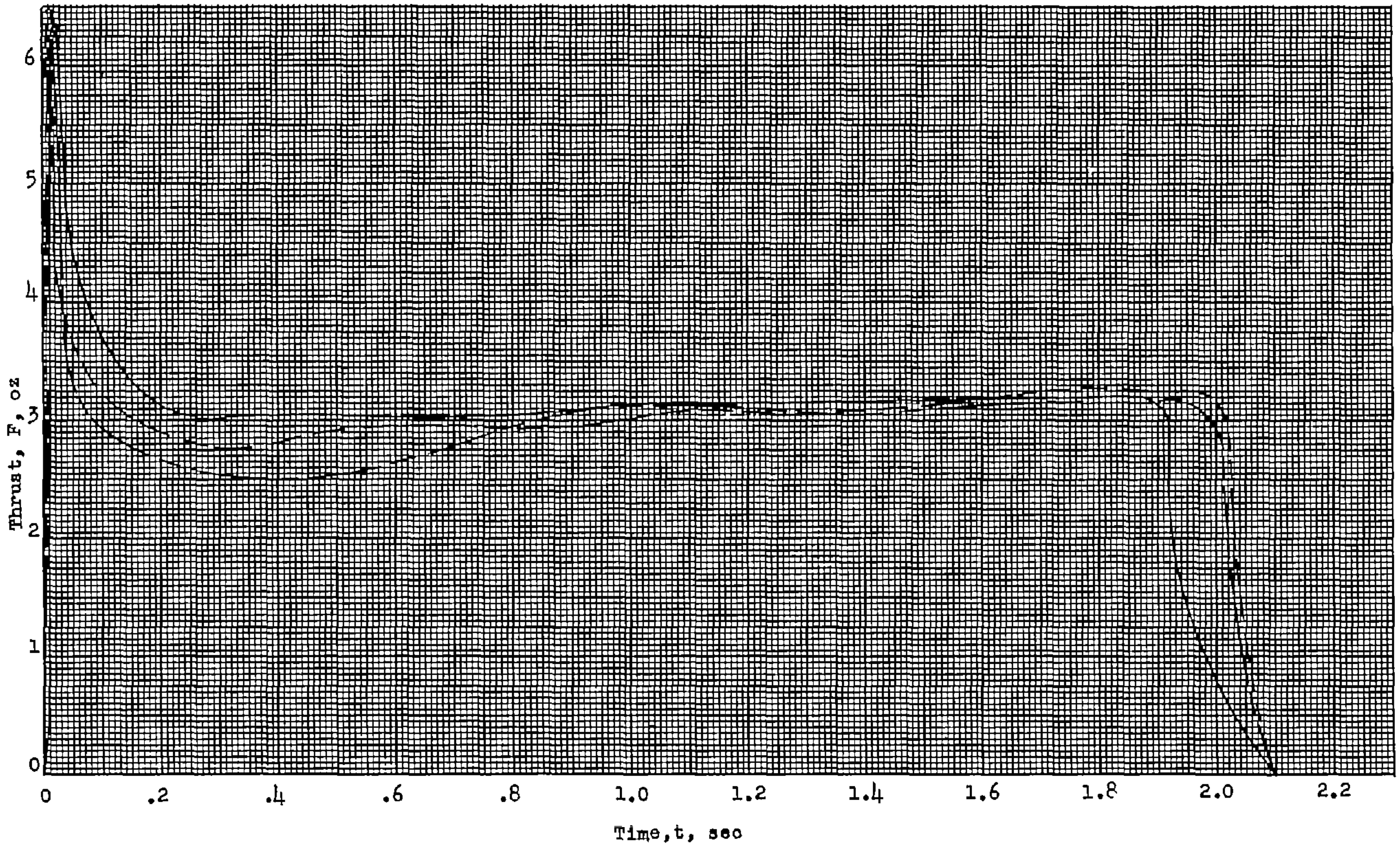

Figure 4.- Variation of thrust with time for three firings of the miniature rocket showing repeatability tests.

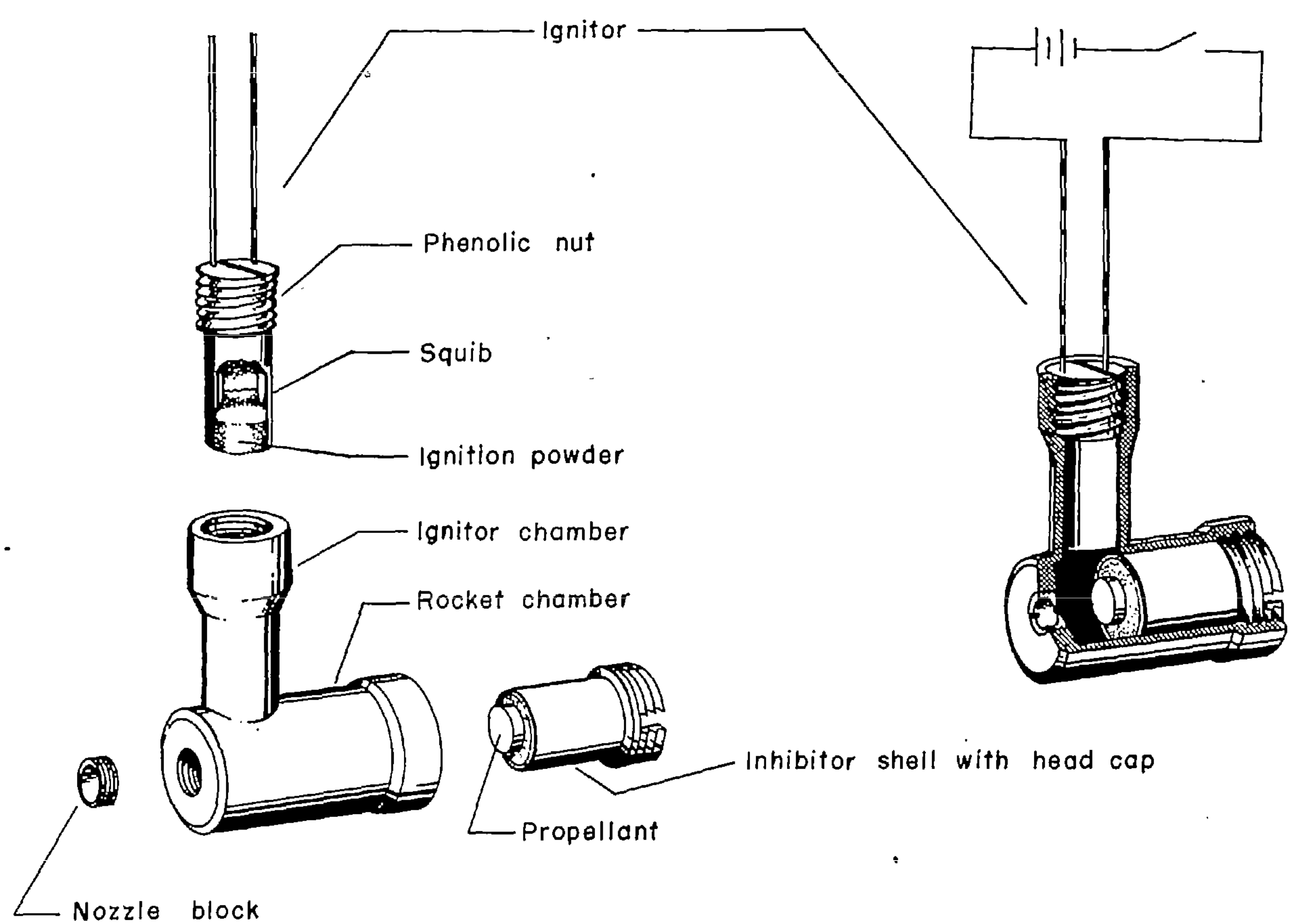

Figure 5.- Sketch of miniature rocket showing the various components and firing circuit.